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PURPOSES AND METHODS OF GAS TURBINE COOLING

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Abstract

The use of turbine cooling permits increasing turbine inlet temperatures to a point where very large gains in engine performance are possible. In addition, a greater freedom in turbine design is obtained because of higher permissible stress levels and a larger choice of possible turbine materials. The gains to be made in engine performance and the methods that can be used to cool turbines are presented.

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INTRODUCTION

Past history has shown that in order to obtain desirable cycle temperatures for heat engines it was often necessary to cool certain engine components to circumvent material strength limitations. From a thermodynamic point of view removal of heat from a cycle by cooling is detrimental to performance, but from a practical standpoint the cooling obtained has permitted a type of engine operation resulting in engine performance that would have never been possible without cooling. An excellent example is the piston-type engine where refinements in cooling

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methods, necessitating large amounts of heat removal, led to increasingly superior performance in the engines. It can be expected that a similar type of evolution will lead to the use of cooling in gas turbine engines to obtain a type of operation that will result in vast new performance possibilities.

There are three main purposes of cooling the turbines of gas turbine engines. The first, and most commonly accepted purpose, is that higher engine cycle temperatures (resulting in increased specific power) can be obtained if the turbine disk and blade temperatures are made independent of the turbine inlet gas temperature. The second purpose of cooling is to permit the use of higher operating stresses and/or longer turbine life by reducing the operating metal temperature of the turbine. Higher operating stresses will permit a much greater amount of freedom in the turbine design and can generally result in increased power for a given engine frontal area by allowing the flow capacity of the turbine to increase through the use of longer turbine blades. The third purpose of turbine cooling is to permit a greater degree of freedom in the choice of turbine materials than is presently available to engine designers. Materials currently used for gas turbines are chosen for strength characteristics at high operating temperatures. In general, these materials also contain relatively large quantities of scarce or critical alloying elements. By lowering the operating temperature of the turbine other materials are made available that in addition to having superior strength properties also have reduced amounts of critical alloying elements relative to the so-called "high-temperature alloys."

It is the purpose of this paper to present some of the performance possibilities that are obtainable through the use of turbine cooling to permit use of higher cycle temperatures, higher turbine stresses, and less critical materials. In addition some of the progress that has been made in turbine cooling research will be described, and an indication will be given as to the trends to be expected in the future development of cooled gas turbine engines.

POTENTIAL ENGINE PERFORMANCE AT HIGHER TURBINE INLET TEMPERATURES

Turbojet Engines

The thrust of a turbojet engine is essentially proportional to the product of the weight flow of air that passes through the engine and the velocity of the jet at the exhaust nozzle. All methods of increasing the thrust output, therefore, depend upon increasing either one, or both, of these variables in some manner. The weight flow of air per unit frontal area can be increased through the use of recently developed compressors. The jet velocity can be increased through use of (1) higher compressor pressure ratios, higher turbine efficiencies, or lower burner pressure losses, all of which result in a higher available pressure in the exhaust nozzle and consequently will allow higher expansion ratios for obtaining higher velocities, (2) improved exhaust nozzle efficiencies, and (3) increased jet exhaust temperature.

Several of the above-mentioned effects on engine performance are illustrated in figure 1 for both nonafterburning and afterburning turbojet engines operating at supersonic speeds in the stratosphere. With the newer compressors the flow capacity of the engine is often determined by the turbine; therefore the engine diameter is often fixed by the turbine diameter. For this reason the relative engine thrust is given per unit of turbine frontal area in figure 1. For this case high compressor pressure ratios result in higher gas densities at the turbine and consequently increased flow capacity. (The compressor pressure ratios shown are for the actual operating conditions and not sea-level static.)

It can be seen from the figure that for nonafterburning engines, turbine inlet temperature has the most significant effect on thrust, and by a simultaneous increase in turbine inlet temperature and compressor pressure ratio the thrust can be increased by a factor of two or more relative to present engines. At constant compressor pressure ratio increases in turbine inlet temperature generally result in increased specific fuel consumption. This can be explained by the fact that the thrust is increased directly as the jet velocity is increased, but the kinetic energy of the gases is increased as the square of the jet velocity. The fuel consumption is proportional to the kinetic energy increase; therefore the fuel consumption increases at a greater rate than the thrust with a resultant increase in thrust specific fuel consumption. This effect can be largely compensated for by increasing the engine thermal efficiency through use of higher compressor pressure ratios.

For afterburning engines figure 1 shows that very high thrust levels are obtainable by increasing the gas temperature in the jet nozzle. This thrust is obtained at a relatively high cost in specific fuel consumption, particularly at the turbine inlet temperatures of current engines. By increasing the turbine inlet temperature a smaller pressure drop is incurred across the turbine and less fuel is burned in the exhaust nozzle (more is burned ahead of the turbine, though) and the thrust output of the engine is increased at a substantial saving in specific fuel consumption. The advantage of turbine cooling to permit increased turbine inlet temperatures for afterburning engines is, therefore, primarily to decrease fuel consumption. In addition, at higher compressor pressure ratios combined with high turbine inlet temperatures, very large increases in thrust are obtainable at no sacrifice in fuel consumption.

Turboprop Engines

The power output of the turboprop, or other shaft power turbine engines, is primarily a function of energy level of the gases ahead of the turbine; therefore the turbine inlet temperature has a direct bearing on the power. In the turboprop engine the gases are expanded almost completely in order to extract the maximum power with the result that the pressure level and density at the last stage of the turbine is always low. Turbine flow capacity is therefore almost independent of compressor pressure ratio. Because of this fact power per unit turbine frontal area has little significance. Of most interest is power per pound of compressor air, defined as specific horsepower. The effect of turbine inlet temperature and compressor pressure ratio on relative specific horsepower and specific fuel consumption is shown in figure 2. Here, as in figure 1, for the turbojet engine the power output can be increased by a factor of two or more, relative to present engines, by increasing the turbine inlet temperature. It will be noted, however, that opposite to the case for the turbojet, increasing turbine inlet temperature also decreases specific fuel consumption for the turboprop. This trend can be explained by the fact that turbine power is a function of the kinetic energy of the gases. If all components were 100 percent efficient, specific fuel consumption would be independent of gas temperature. With inefficiencies in the components, however, the losses are proportionally greater at low temperatures than at high temperatures; therefore the specific fuel consumption decreases with increasing gas temperature.

HIGHER STRESSES AND MATERIAL FREEDOM OBTAINABLE BY TURBINE COOLING

The flow capacity of modern compressors is rapidly developing to the point that the components aft of the compressor (primary burner, turbine, afterburner, and exhaust nozzle) will determine the engine frontal area. For many applications it appears that the turbine may be the largest diameter component of the engine. The flow area of the turbine can be increased by use of longer turbine blades, but this often results in stresses in excess of those permissible using presently available materials in uncooled turbines. An indication of how turbine diameter is related to turbine blade stress for a given compressor weight flow and pressure ratio is shown in figure 3.

Increasing the blade root stress from 30,000 to 60,000 pounds per square inch can result in a reduction in turbine diameter of almost 15 percent. This corresponds to a frontal area reduction of over 25 percent.

Methods of obtaining higher turbine stress levels are indicated in figure 4. For the temperatures at which gas turbine blades operate stress-rupture is the criterion that usually determines allowable blade stress. The stress-rupture properties of several materials are shown in figure 4 and the upper levels of the curves are cut off where stress-rupture properties no longer determine the permissible stress. The alloy S-816 is a commonly used alloy in present gas turbine engines. At a temperature of 1500° F (about standard blade temperature for present engines) the maximum allowable stress is shown to be 24,000 pounds per square inch. If, however, the temperature is reduced only 100° F by cooling, the allowable stress can be increased by about 35 percent, with further increases being obtainable at lower temperatures. Below temperatures of about 1200° F, however, other materials, such as A-286, possess better strength properties with the possibility of operating at stresses over 90,000 pounds per square inch - over $3\frac{1}{2}$ times the allowable stress for present engines. Further reduction in temperature makes it possible to utilize high strength steels such as the Timken alloy 17-22A(S). Use of this type of material offers only slight increases in possible operating stress relative to A-286, but the critical material content of 17-22A(S) is almost completely eliminated. This alloy is about 97 percent iron. Even the alloy A-286 is presently considered to be a relatively noncritical alloy because it is over 50 percent iron and contains no cobalt or columbium. Currently used blade materials such as the alloy S-816 contain very high quantities of critical materials such as cobalt, nickel, chromium, and columbium - the iron content of S-816 is only 2.8 percent.

In general, the higher strength materials at temperatures from 1000° to 1300° F all contain considerably smaller quantities of critical materials than the currently used high temperature alloys. The significance of this fact is that in addition to cooling permitting the use of higher turbine inlet temperatures and stresses in turbine engines, it also permits the production of a greater number of engines due to the greater availability of suitable materials for the turbines. This last effect of availability was the primary reason that the Germans incorporated the use of turbine cooling in some of their turbojet engines in World War II.

EFFECT OF TURBINE COOLING ON ENGINE PERFORMANCE

The most probable source of air for turbine cooling purposes is the engine compressor. It may often be possible to bleed the compressor from some intermediate stage in order to keep the compressor work on the cooling air to a minimum, but for turbine rotor cooling it is expected that more often it will be necessary to bleed from the discharge of the compressor in order to obtain as high a pressure as possible for effective blade cooling. The use of this air for cooling has several effects on engine performance. The air is removed from part of the engine cycle so that it is unavailable for doing work in the turbine, but work is done on the cooling air which makes the specific turbine work higher than without bleed. When the air is used for turbine rotor cooling additional work is done on the cooling air to accelerate it to the wheel speed at the blade tip as it passes through the rotor. For a given value of turbine inlet temperature the pressure ratio across the turbine will therefore be somewhat higher when air is bled from the compressor. For engines utilizing jet thrust some of the cooling air energy can be recovered in the exhaust jet by adding weight flow to the jet.

Based on present knowledge of the quantities of cooling air that will be required for operation at higher turbine inlet temperatures, it is possible to predict the engine performance attainable utilizing air-cooled turbines. Turbojet engine performance with no consideration given to the effects of bleeding part of the compressor air for turbine cooling was shown in figure 1. Since materials are not available that can withstand very high gas temperatures without cooling, an incremental decrease in ultimate uncooled engine performance must be tolerated in order to reap the benefit of gains made possible through use of higher gas temperatures by turbine cooling. In figure 5 the expected air-cooled engine performance obtainable using the better methods of cooling presently under development is shown. (The quantity of air required for cooling varies with turbine inlet temperature.) It can be noted that cooling generally results in a small decrease in thrust with practically no sacrifice in fuel consumption relative to the calculated performance without cooling for nonafterburning engines. In afterburning engines the effect of cooling results in increased fuel consumption with only small effects in thrust. These effects can be explained by the fact that cooling generally shifts the performance map in the direction of lower turbine inlet temperatures by diluting the exhaust gases and lowering the temperature after the turbine.

The performance of an air-cooled turboprop engine at sea-level static conditions is shown in figure 6. The decrease in performance relative to the calculated performance without cooling (from figure 2) is greater for the turboprop engine than it is for the turbojet engine due to the fact that in the turboprop the cooling air passing through the cooled portion of the turbine is completely lost for engine power generation in that portion of the turbine. In the turbojet engine, on the other hand, the cooling air is still available for obtaining power in the form of jet thrust.

Also shown in figure 6 is a point showing performance for a liquid-cooled turbine for the most severe conditions shown. In this case the only effect of cooling on the cycle is removal of a small portion of the heat from the gases. The effect on performance is extremely small. From a performance standpoint liquid cooling of turboprop engines is very promising providing radiator drag losses are tolerable. Even though the performance of air-cooled turboprop engines is inferior to the performance for liquid-cooled engines, the use of air cooling is still very promising due to the fact that substantial increases in power are obtainable at no increase in specific fuel consumption relative to uncooled engines at current gas temperature levels.

The effects of bleeding various amounts of air from the compressor for cooling or other purposes is shown in figures 7 and 8 for turbojet and turboprop engines. The quantity of air required for cooling is dependent upon the type of air-cooled blades used in the engine; therefore at given engine conditions a wide variation in cooling-air requirements is possible. In most aircraft gas turbine engines air is bled from the compressor for such uses as accessory drives or cabin cooling. In this case the air is thrown overboard and cannot be used for jet thrust or turbine power. As a basis of comparison the effects on performance of this prevalent practice of overboard bleed is shown in addition to the effects of bleeding air for turbine cooling purposes. The relative specific thrust and specific fuel consumption variations with various percentages of cooling air bled from the compressor discharge for a turbojet engine flying at supersonic speeds in the stratosphere are shown in figure 7. The same general trends are obtained at conditions other than those given for this figure. The thrust decreases approximately 1 percent for every percent of air bled from the compressor for turbine cooling purposes. This loss is more than doubled if the air is bled overboard so that it cannot be used for jet thrust. The effect of air bled for turbine cooling has only a very slight effect on specific fuel consumption, but overboard bleed has a very large effect.

The effect of bleeding various amounts of cooling air on the performance of a turboprop or stationary power plant operating at sea-level static conditions (no jet thrust horsepower) is shown in figure 8. As explained in connection with figure 6 the incremental decrease in performance is larger for a turboprop engine than for a turbojet engine. This is particularly true concerning specific fuel consumption where the increase with cooling air flow is more than 5 times as high as for a turbojet engine. The decrease in power due to bleeding air for turbine cooling is only slightly higher than decrease in thrust for a turbojet. For either turbojet or turboprop engines it is less costly to bleed air for turbine cooling purposes than it is for cabin cooling, accessory drives, electronic equipment cooling, etc.

The effect of liquid cooling on turboprop performance is extremely small. The change in power or specific fuel consumption due to liquid cooling at the conditions of figure 8 is less than 1/2 of 1 percent.

As explained in reference 1 there is an optimum jet velocity for turboprop engines at each flight speed that results in minimum fuel consumption and maximum thrust. The incremental decreases in performance due to bleeding air from the compressor for cooling purposes are somewhat smaller for turboprop engine with optimum jet thrust than for the case with no jet thrust, but the improvement is of little significance.

Results shown in figures 6 and 8 indicate that for turboprop engines liquid cooling would be better than air cooling. Substantial improvements in engine performance are obtainable, however, with either liquid or air cooling being used to permit engine operation at higher turbine inlet temperatures.

METHODS OF TURBINE BLADE COOLING

Both air and liquid cooling methods have their relative advantages and disadvantages as will be briefly discussed later. Air cooling appears to have the most promise for turbojet engines. Because the scope of this paper is limited by space considerations, the discussion of methods of cooling will be limited to air-cooling.

Incorporation of air cooling into an engine will affect the whole engine design in order to make the most effective use of the air. A possible engine configuration is shown in figure 9. Air is ducted from the discharge, or one of the late stages, of the compressor to the turbine rotor and stator. The air is then discharged into the gas stream downstream of the turbine to provide additional thrust. A hollow turbine shaft provides a very convenient method of ducting cooling air to the rotor and it can often eliminate many air seal problems.

Various types of air-cooling methods are shown in figure 10. The most conventional method of cooling used in all heat-transfer processes is convection cooling, illustrated in figure 10(a). In order to augment the cooling effectiveness by this means, it is desirable to add heat-transfer surface area on the heat rejection side of the apparatus in the form of increased surface such as the fins shown. This is the method of cooling that has been successfully used on air-cooled piston-type engines for years. The next method of cooling shown (fig. 10(b)) is less well known. With this method a film of cool air is introducted through slots to form an insulating layer between the hot gases and the cooled surface. Air has very low thermal conductivity so that it is a good insulation medium, but the effectiveness of the layer of air is lost some distance downstream from the slot by mixing with the hot gases. This disadvantage is eliminated by transpiration cooling (fig. 10(c)) because air is bled through a porous surface continually over the entire area of the surface. Transpiration cooling is the most effective method of air cooling known at the present time. A comparison of the cooling effectiveness of these three methods of cooling (convection, film, and transpiration) is given in reference 2.

Air-cooled turbine blades utilizing these various methods of cooling are illustrated in figure 11. The hollow blade was used by the Germans in some of their turbojet engines in 1945, but the cooling effectiveness of this type of blade is so low that excessive quantities of cooling air are required. The tube-filled blade was an early attempt of the NACA to provide extra surface area inside of the coolant passage in order to improve the blade cooling effectiveness. Some results of tests and methods of manufacture on this type of blade are given in references 3 through 7. Much valuable information has been obtained from this configuration. In order to try to improve the cooling effectiveness in the leading and trailing edge regions, which are difficult to cool, film cooling was investigated on the type of blade shown in figure 11(c) and reported in references 8 to 10. Cooling in these regions was effective, but blade durability was found to be a serious problem (ref. 6). Another solution to the problem of leading and trailing edge cooling is to increase the thermal conductivity of the blade shell; consequently copper-clad shells as shown (fig. 11(d)) were investigated in reference 9. This type of structure is similar to copper-clad kitchen utensils that are used to spread the heat over the entire area of the utensil. The biggest disadvantage of the copper-clad blade is that the weight of the blade is increased by the copper so that the stress is increased to a point where the gains in cooling are practically eliminated.

A practical type of shell-supported convection cooled blade construction is the corrugated blade (fig. 11(e)). Large amounts of heat-transfer surface area can be added in the form of corrugated fins and the fins can be made to extend well into the leading and trailing edges of the blade to insure adequate cooling in these regions. An island is usually provided in the middle of the passage so that the corrugations can be of uniform amplitude. This island is blocked off so that no cooling air passes through this region. Temperature data obtained from this type of blade will be discussed subsequently.

In all of the turbine blades discussed up to this point, the blade shell has been the primary support member for carrying the stresses due to centrifugal forces. Because of the fact that the shell is exposed to the gas stream, it is also the hottest member of the blade and therefore the stress-carrying capacity is lower than for portions of the blade that are cooler. Blades have therefore been designed and tested where the main stress-carrying member, or strut, is submerged inside of the coolant passage and operates at a lower temperature than the blade shell. The shell can be made thin and can be completely supported by the strut. In this manner the stresses in the shell are greatly reduced with the result that it can operate at higher temperatures. With higher shell temperatures the heat transfer from the gas to the blade is reduced and the quantity of cooling air required is reduced. This type of blade shows great promise for future application in air-cooled turbine engines.

The last blade shown in figure 11 is a transpiration-cooled blade. The porous shell could be made from several materials - the most probable materials are woven wire cloth or porous sintered materials made from powered metal. Only a limited amount of unclassified experimental data is presently available for transpiration-cooled turbine blades. Some of the early results are given in references 11 and 12. Some of the advantages and problems in the use of transpiration cooling are discussed in reference 13.

Experimentally measured turbine blade temperatures are shown in figure 12 for the corrugated blade illustrated in the inset. The "coolant-flow ratio" used as the abscissa in the figure is defined as the ratio of the air used for turbine cooling purposes to the total flow of air through the compressor. It will be noted that for the uncooled condition the blade temperature is over 200° F lower than the turbine inlet temperature. This difference is due to the fact that, because of high rotative speeds of the turbine and high gas velocities at the stator exit, the total temperature relative to the turbine rotor blades is less than the total temperature relative to the stator blades. By the use of only 2 percent of the compressor air for turbine rotor cooling purposes, the blade temperature can be reduced over 400° F below that of an uncooled blade, or approximately 650° F below the turbine inlet temperature. These results show the substantial blade temperature reductions that are possible with very nominal amounts of cooling air. In addition it is shown that very good success has been obtained in developing methods of predicting the average blade temperature of this type of blade. The variation between measured and predicted temperatures is less than 35° F. The predictions were based essentially on the methods discussed in references 14 and 15. The success of the analytical methods of predicting blade temperatures gives encouragement for using these methods for predicting cooling requirements for other conditions.

A comparison of the cooling effectiveness of the corrugated blade with the strut-supported blade is shown in figure 13. In most cases the coolant flow required for the strut-supported blade is about one-half that required for the corrugated blade in order to obtain a specified blade temperature. At the very low flow rates this difference in flow will have very little significance in the over-all engine performance, but as the cooling load becomes more severe, the results of figures 7 and 8 show that the savings in cooling air using the strut-supported blade could result in appreciable gains in engine performance. Also shown in figure 13 is the comparison between experimental and calculated strut temperatures. Again the agreement is very good. The calculated temperatures were obtained by the methods described in reference 16.

A word of caution is required concerning generalization of the results obtained from research on the blades shown in figures 11 through 13. All of these blades had a chord of approximately 2 inches or more. A blade of this size is satisfactory for many applications. For some engines, such as most turboprop engines, however, the blades are much smaller and it is not always possible to scale down the designs shown. Further research is required on small air-cooled turbine blades.

Besides the temperature reduction that is obtained by cooling, air-cooled blade durability is also of importance. Endurance investigations have been conducted, mostly on tube-filled blades, and are partially reported in reference 6. It has been demonstrated that endurance life and reliability can be at least as good, if not better, than for uncooled turbine blades. Air-cooled turbine blade lives in excess of 300 hours have been obtained with no indication of failure. There is no reason to believe that the type of construction required for air-cooled turbine blade should have a detrimental effect on blade life.

AIR-COOLED TURBINE DISK CONFIGURATIONS

The use of air-cooled turbine blades will require a type of turbine rotor construction different from current practice. There is, however, a considerable amount of freedom in the type of design possible, and up to the present time experience has not shown a marked superiority of any particular type of design. Various possible types of turbine disk construction are shown in figures 14 and 15. Two main types of construction are the split disk (fig. 14) and the shrouded disk (fig. 15). With either type of construction the cooling air can be supplied from the upstream direction, the downstream direction, or through a hollow turbine shaft. With any of the types of construction internal vanes are required in the turbine rotor in order to direct and help pump the cooling air out to the blades. Vane configurations for two experimental split-disk, air-cooled turbines are shown in figure 16. In one case the vanes were curved to provide an inducer section for the cooling air, while in the other case straight vanes were used. Experimental tests have not indicated a superiority for either type of vane construction.

Up to the present time experimental tests have been conducted only on the disk configuration shown in figure 14(b) and some of the results are presented in references 17 through 19 which indicate that cooling of the disks will be adequate using the amount of air required for blade cooling. Several turbines have been built and the downstream inlet was required in order to minimize the alterations to a commercial engine for incorporation of turbine cooling.

PROBABLE FUTURE USE OF VARIOUS TURBINE COOLING METHODS

Most experimental research up to the present time on turbine cooling has been concerned with investigating various possible types of blade configurations to determine how well they cool, fabrication problems, and expected durability. Investigations have been conducted in commercial engines that were modified to accommodate air-cooled turbines and the turbines were usually made of nonstrategic materials. By a combination of this experimental research and analysis it has been possible to verify analytical procedures, and cycle calculations have been made to determine the areas of operation where future use of turbine cooling will be most profitable.

The use of cooling is particularly promising for turbojet engines powering aircraft at supersonic speeds and in turboprop, or other shaft power turbine engines, for subsonic and transonic flight and for stationary or marine power plants.

The relative merits of three types of blade cooling methods (convection, film, and transpiration) for future use in cooled turbojet engines are indicated in figure 17. The relative coolant flows required for these various types of cooling are shown for ranges of flight Mach number and turbine inlet temperature. In all cases these calculations are based on results in reference 2 and the results are for compressor discharge air being used as the coolant.

At low flight speeds the quantity of coolant required for turbine inlet temperatures up to 2000° F (almost 400° F above current practice) is relatively small for either convection or transpiration cooling. Film cooling does not appear to be practical mainly because blade durability is a problem, but in addition the flow requirements are also higher than for other methods. The convection cooling is representative for the better shell-supported blades (see fig. 11(e)). Strut-supported blade cooling-air requirements would be intermediate between the convection and transpiration cooling.

As turbine inlet temperature is increased to 3000° F at low flight speeds, the cooling requirements become more severe and film cooling appears to be completely out of question. Convection cooling is quite possible, but relative to transpiration cooling the cooling requirements are high. A convection-cooled strut-supported blade would probably be satisfactory.

As flight speeds are increased, the ram-air temperature increases considerably with the result that the temperature of the compressor discharge air, which is used for turbine cooling, rises rapidly. The high cooling-air temperature makes turbine cooling more of a problem at high flight speeds. As shown in figure 17 at a flight Mach number of 2.5 the blade cooling problems are just as severe at a turbine inlet temperature of 2000° F as they were at low flight speeds at a turbine inlet temperature of 3000° F. Increasing the turbine inlet temperature to 3000° F at a flight Mach number of 2.5 makes the cooling problem so severe using compressor discharge air that transpiration cooling is the only air-cooling method left that shows a possibility of efficient operation. It is possible, however, to provide a certain amount of refrigeration to the turbine cooling air. When this is done, the problems of turbine cooling at high flight speeds become less difficult. At flight Mach numbers up to somewhat over 2.0 cooling-air refrigeration is probably unnecessary.

It can be concluded from this and other studies that for gas temperatures up to about 2500° F and flight Mach numbers up to at least 2.0 convection-cooled blades of the corrugated and strut-supported type will probably be adequate. At higher temperatures and higher flight speeds transpiration cooling may be required.

There is also the question concerning the relative merits of air and liquid cooling. Because of its heat-transfer properties water is one of the best liquid coolants and will probably find use in almost any type of liquid cooling system. In most cases the heat picked up by the water will be rejected to ram air. At Mach numbers in excess of 2.0 the ram-air temperature is higher than the boiling point of water at reasonable pressures in the radiator; consequently, heat rejection would be difficult. In addition, water cooling systems are more vulnerable to enemy action in military aircraft so that its reliability is somewhat more questionable. Cooling losses are always less (neglecting effects of radiator drag) with water cooling than with air cooling, but for turbojet engines the difference in performance probably is not of very large importance. It is expected therefore that air cooling will be used for practically all turbojet applications.

For turboprop applications air-cooling losses are much larger than they are for turbojets, but substantial performance improvements are possible using either air or liquid cooling to permit higher turbine inlet temperatures. In addition the flight Mach numbers for turboprop aircraft will probably be always subsonic or transonic so that heat rejection with liquid-cooling systems will not be a serious problem. For aircraft application the better performance of liquid-cooled systems will have to be balanced against less vulnerbility of the air-cooled systems. Either type of cooling system can probably be utilized satisfactorily. For stationary or marine power plants, however, there appear to be no particular advantages of air cooling relative to liquid cooling to permit high temperature operation; consequently, liquid cooling will probably be most satisfactory because it will result in lower fuel consumption rates. For either air or liquid cooling the potential gains in engine performance through use of higher turbine inlet temperatures for both shaft and jet power turbine engines are well worth the effort required to build turbine cooling into the engine design.

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Figure 3. - High turbine stresses result in smaller turbines.

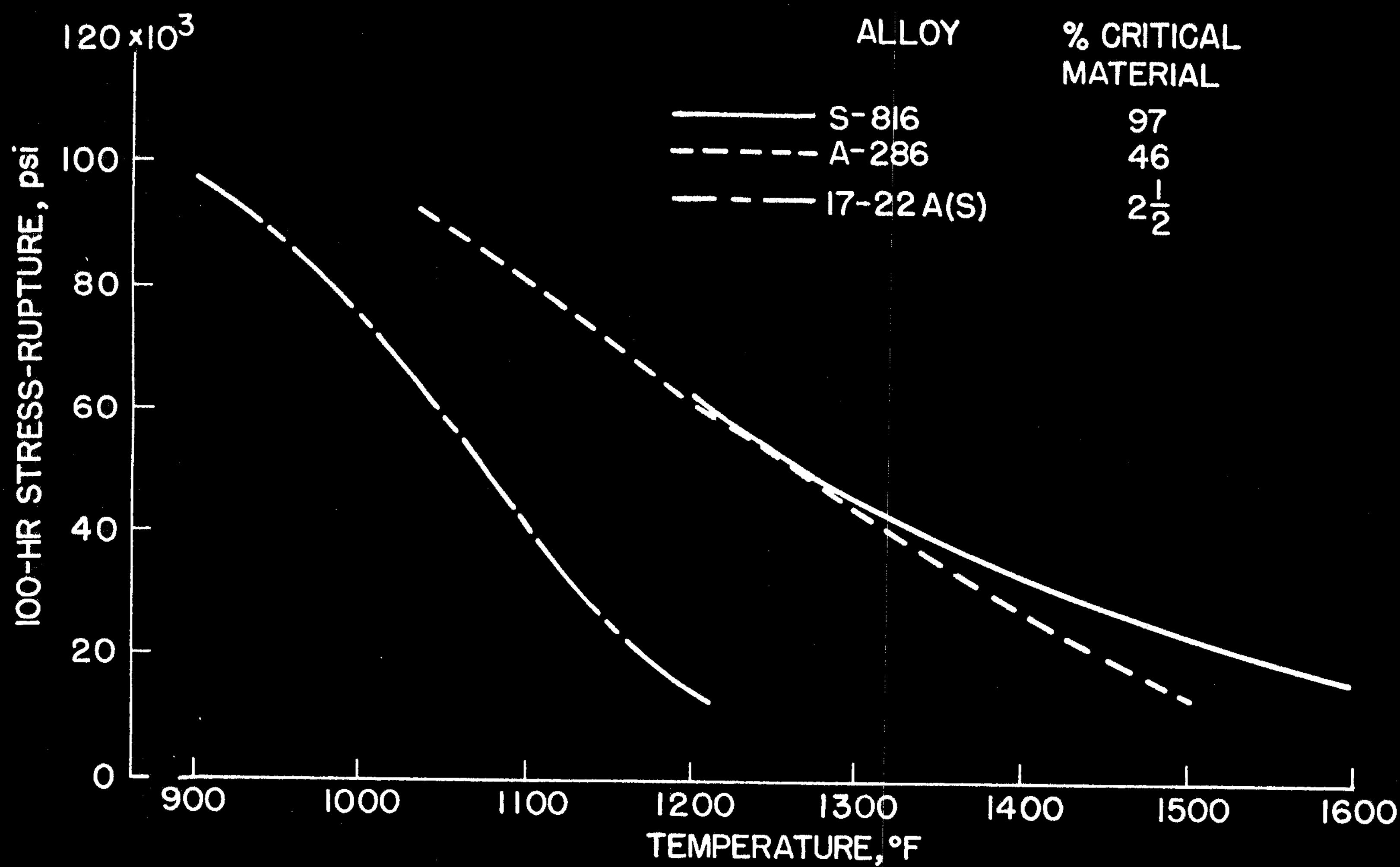

Figure 4. - Stress-rupture properties for possible air-cooled turbine blade materials.

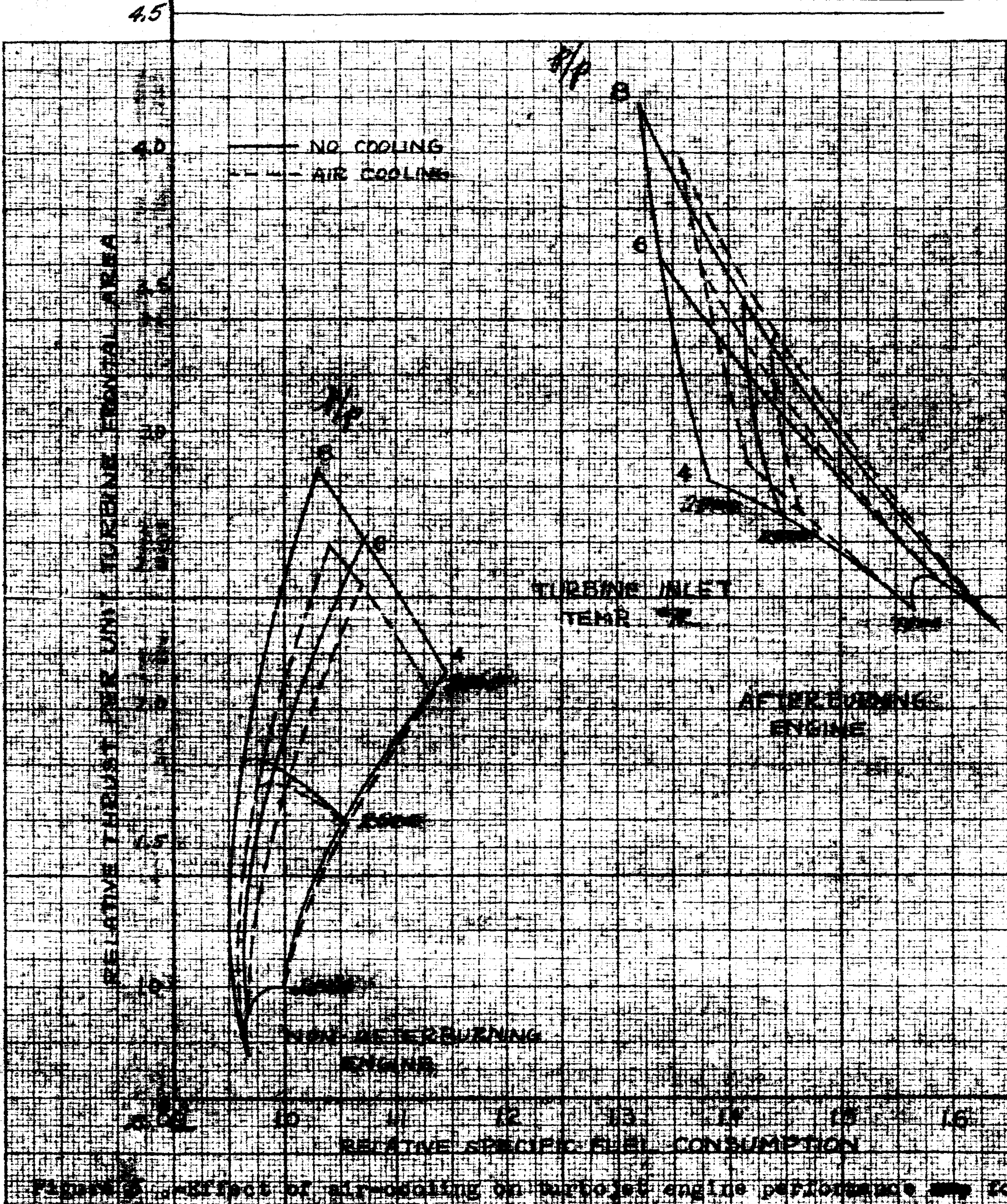

Figure 9. - Effect of air-cooling on turbojet engine performance map for ranges of turbine inlet temperature and compressor pressure ratio at supersonic speeds. Turbine hub-tip radius ratio = 0.7; afterburner gas temperature = 3000 °F.

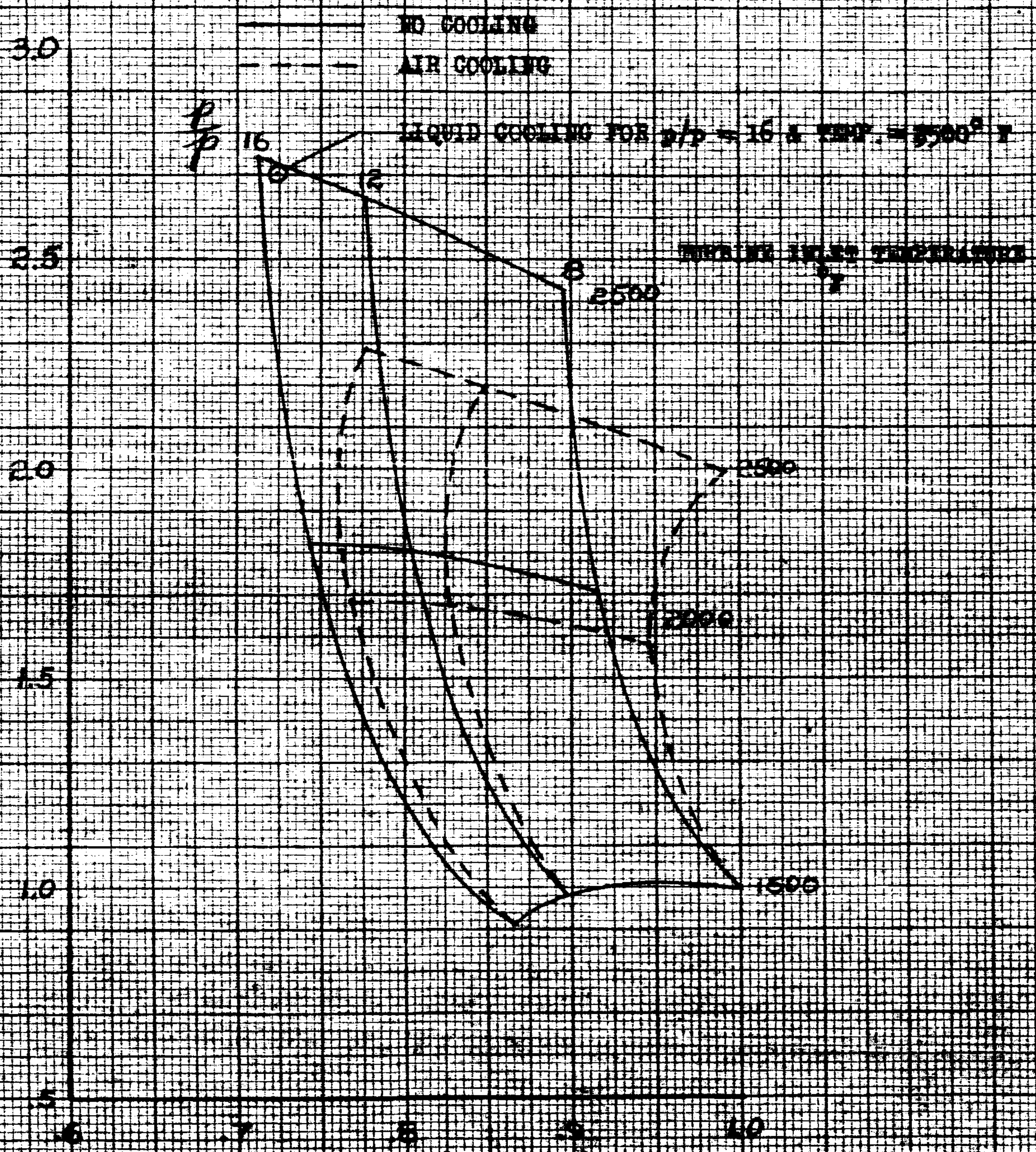

FIGURE 6. EFFECT OF AIR AND LIQUID COOLING ON TURBOPROP ENGINE PERFORMANCE FOR RANGES OF TURBINE INLET TEMPERATURE AND COMPRESSOR PRESSURE RATIO AT SEA-LEVEL STATIC CONDITIONS

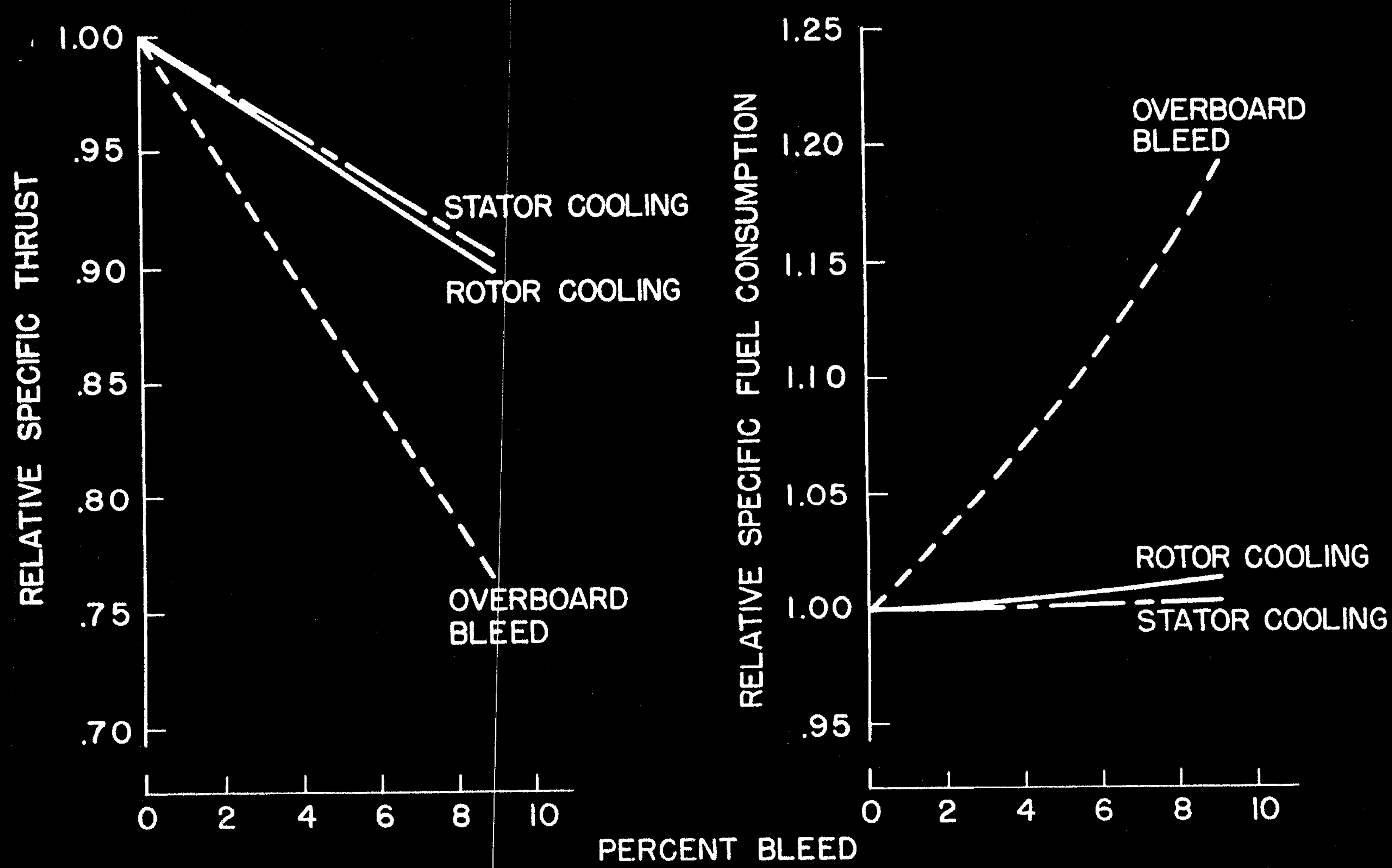

Figure 7. - Variation in turbojet engine performance with various types of compressor discharge air bleed. Turbine inlet temperature, 2000° F; compressor pressure ratio, 6; supersonic flight speed in the stratosphere.

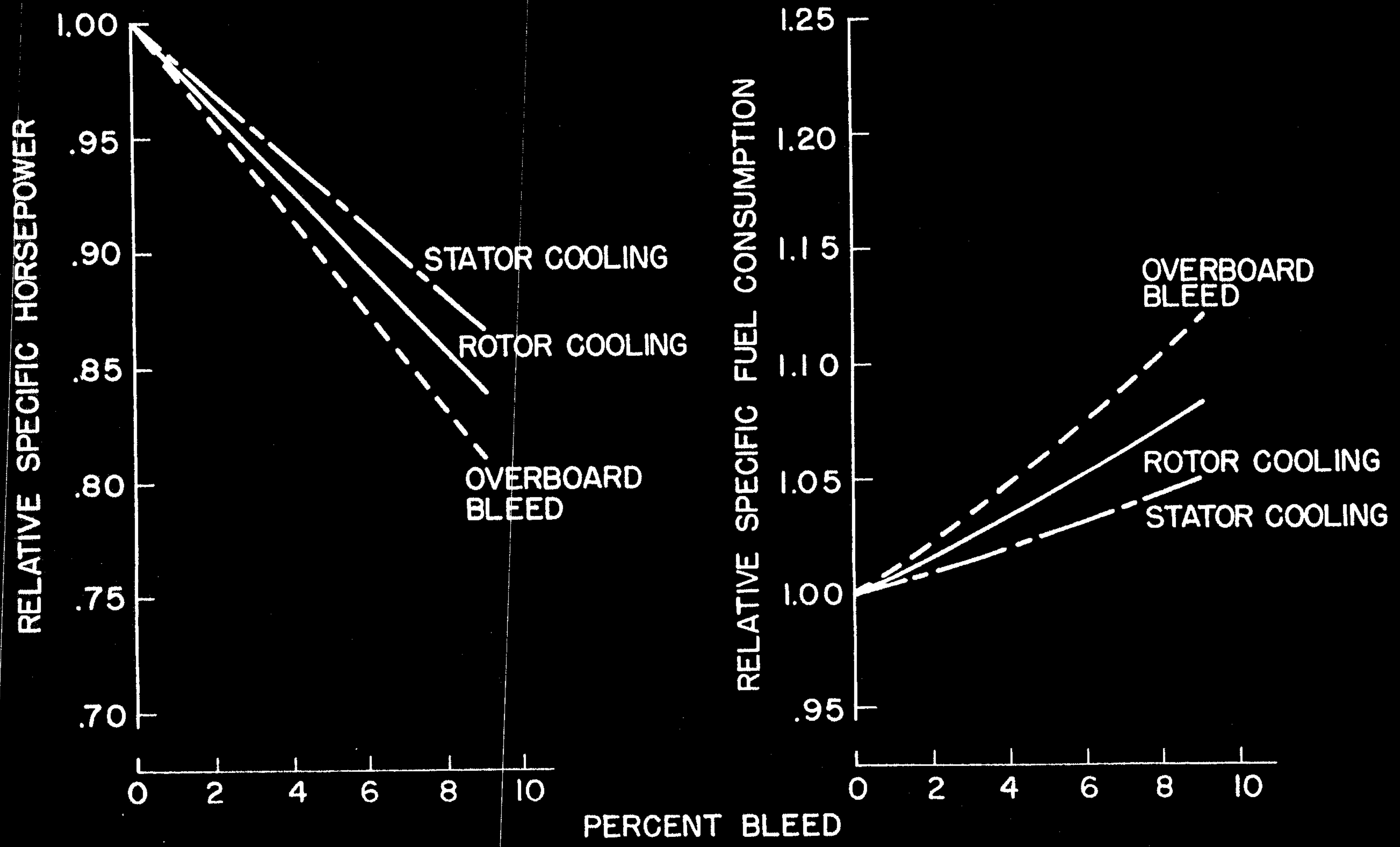

Figure 8. - Variation in turboprop engine performance with various types of compressor discharge air bleed. Turbine inlet temperature, 2000° F; compressor pressure ratio, 12; sea-level static conditions.

75%
100%
C-5

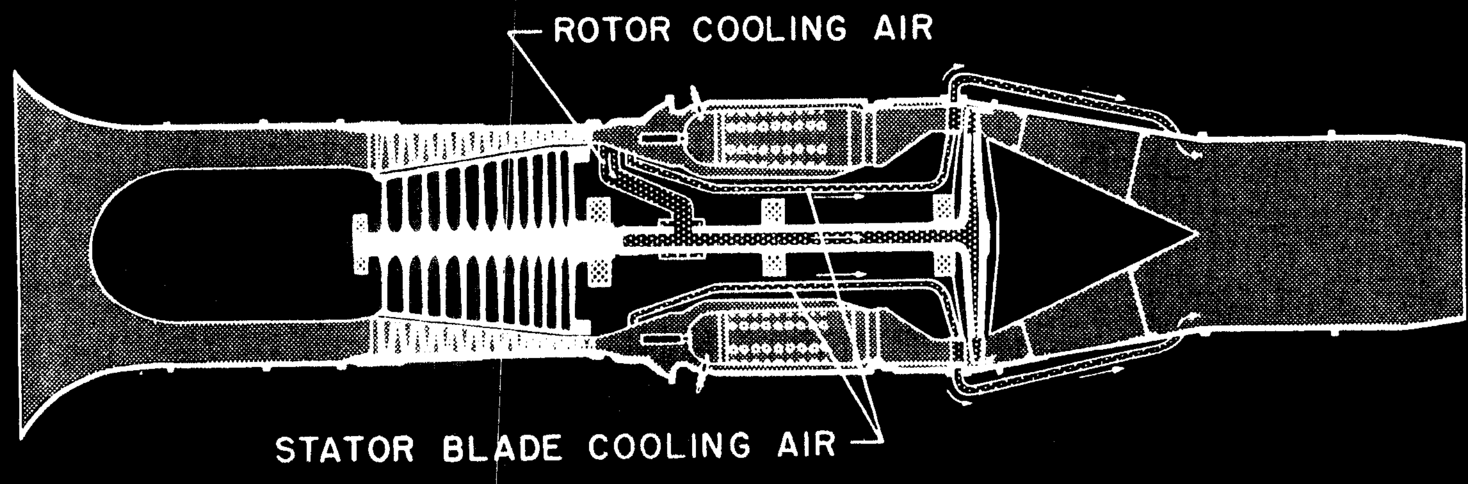

Figure 9. - Cross section of air-cooled turbojet engine.

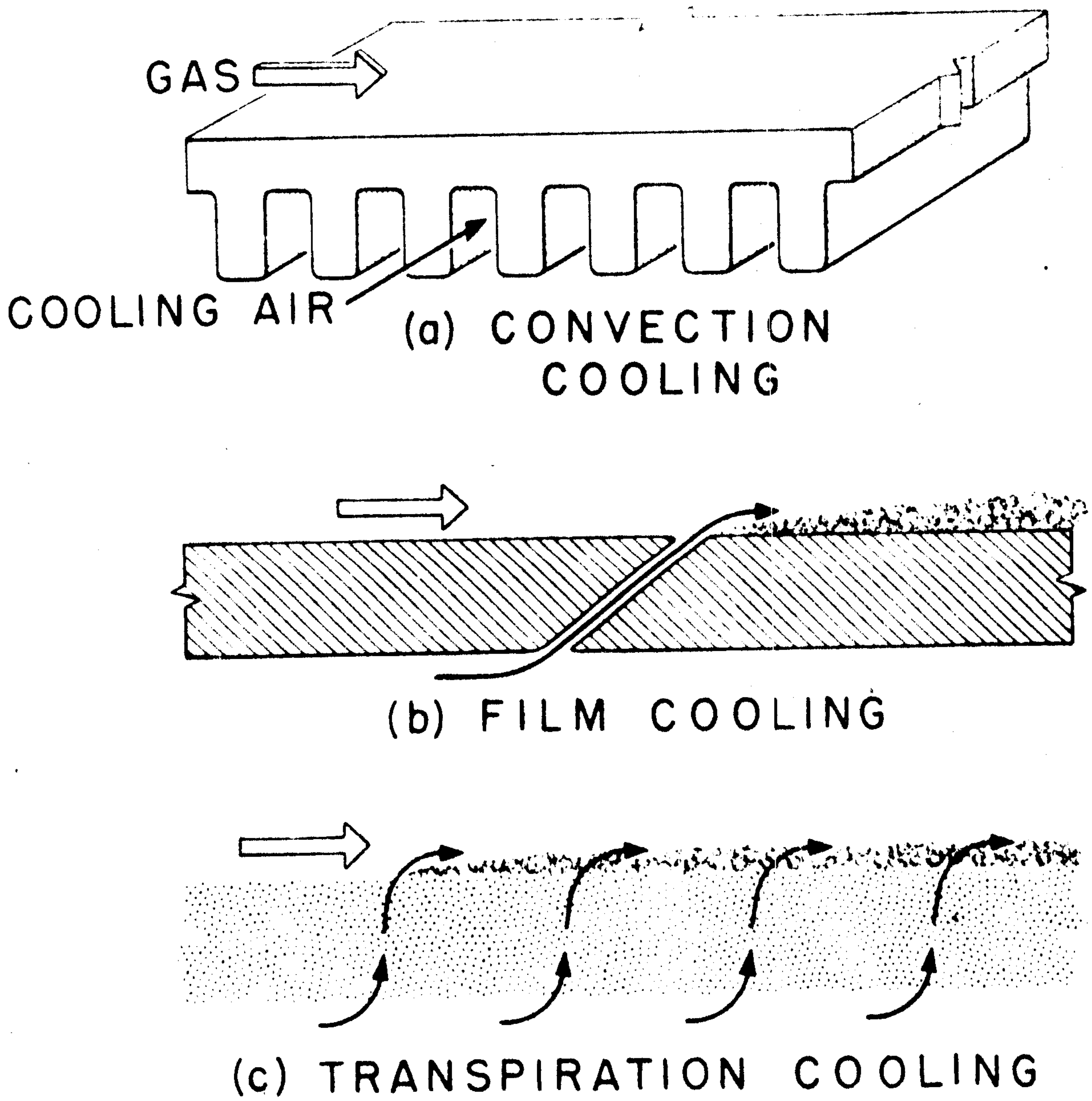

Figure 10. - Three methods of air cooling.

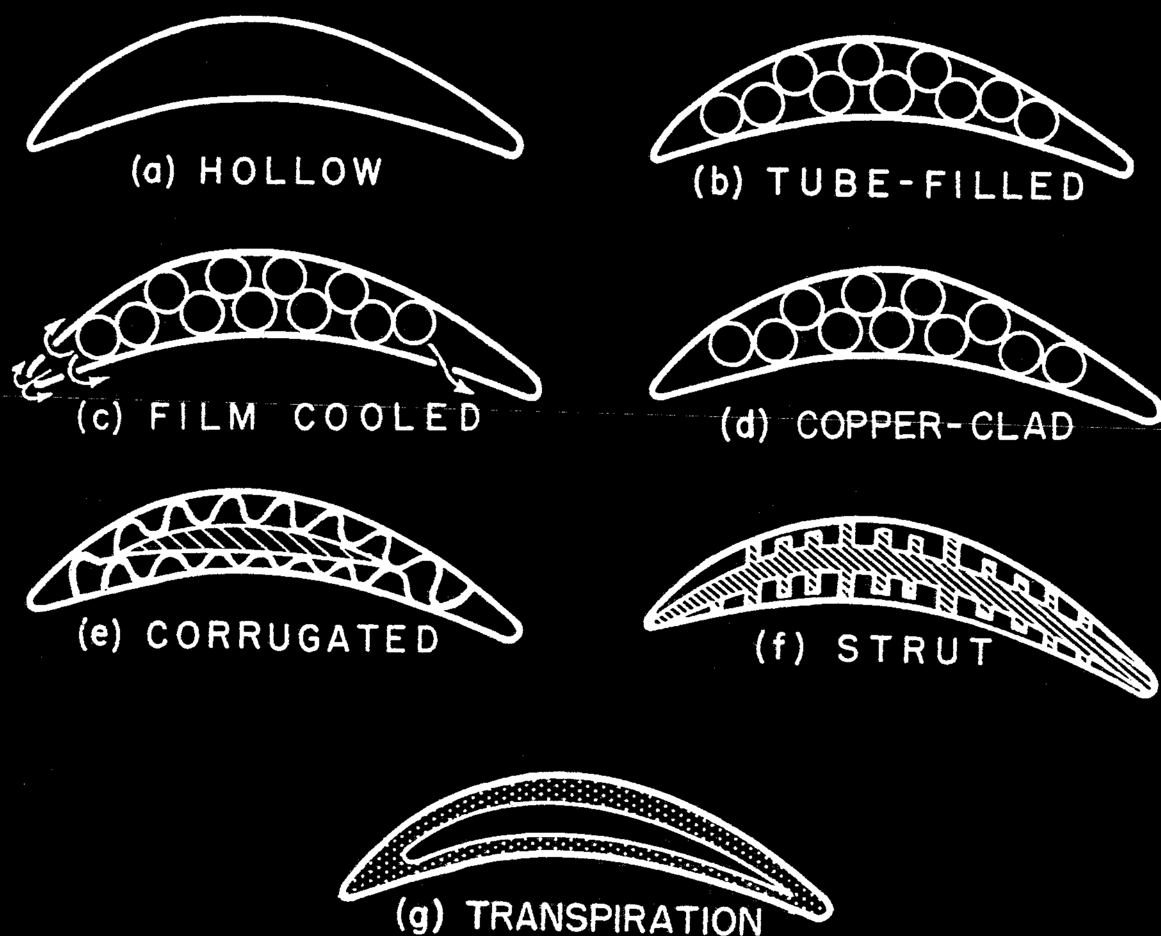

Figure 11. - Air-cooled blade configurations.

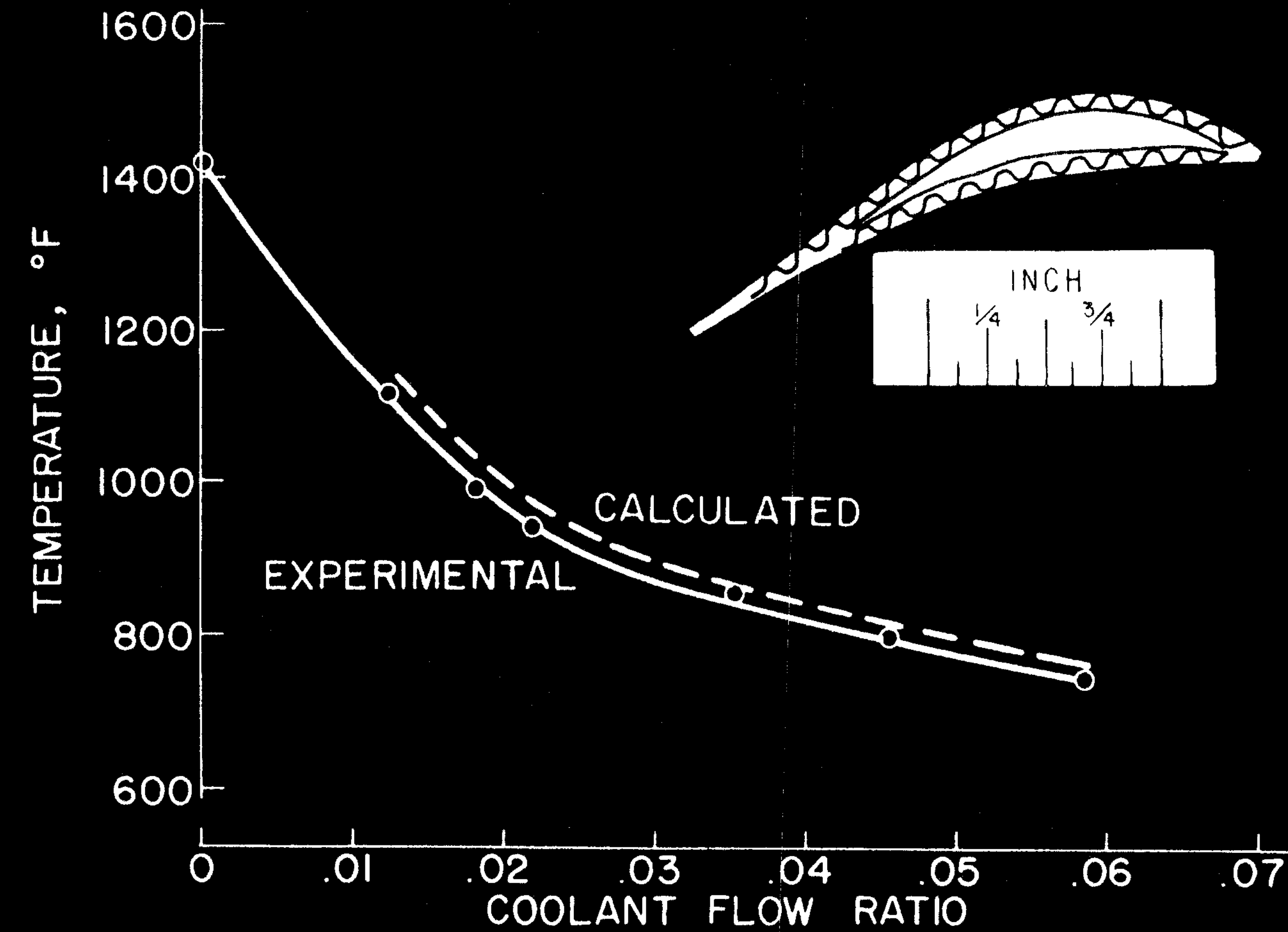

Figure 12. - Comparison of experimental and analytical blade temperatures. Turbine inlet temperature 1640° F.

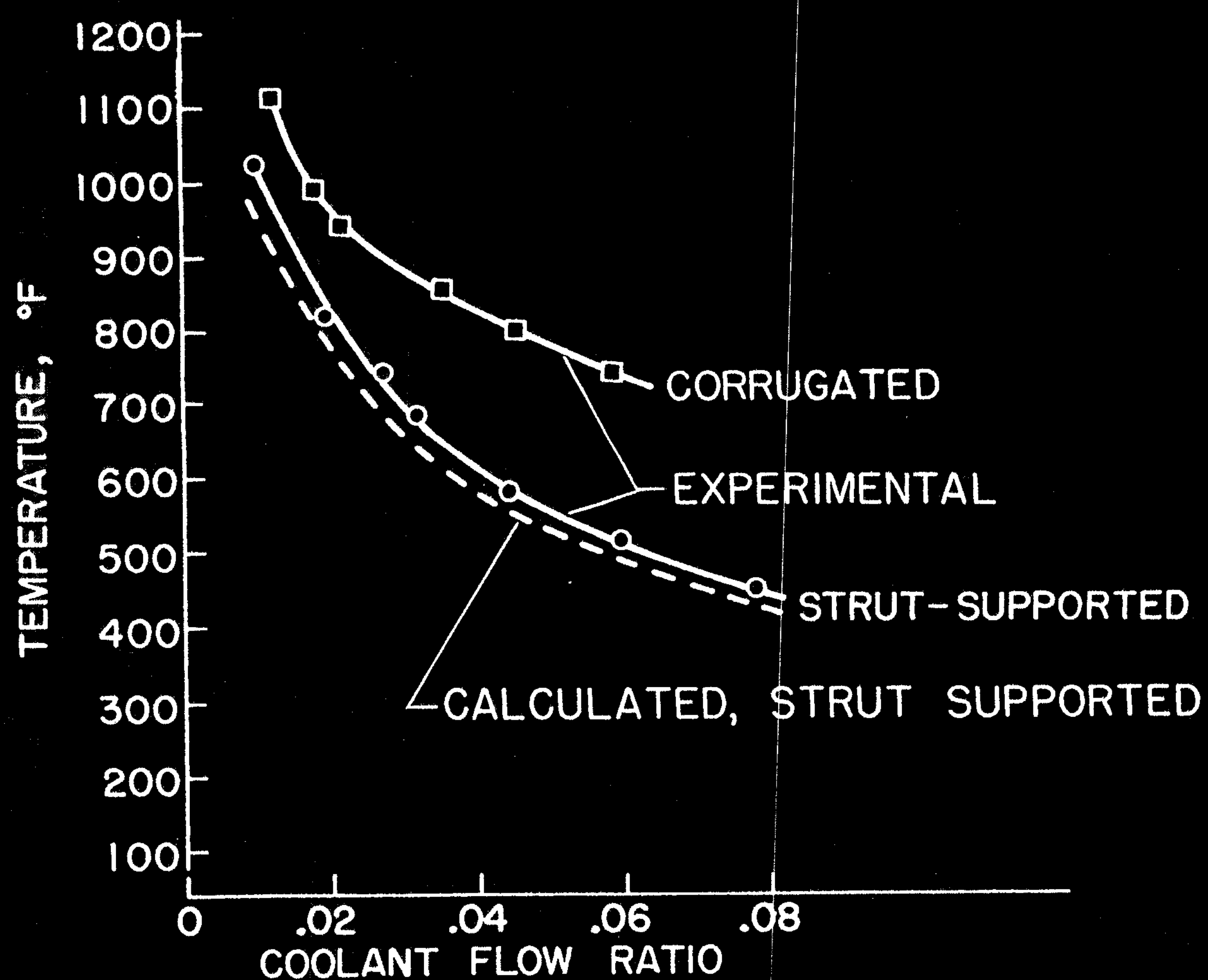

Figure 13. - Temperatures of two convection-cooled turbine blades. Turbine inlet temperature 1640° F.

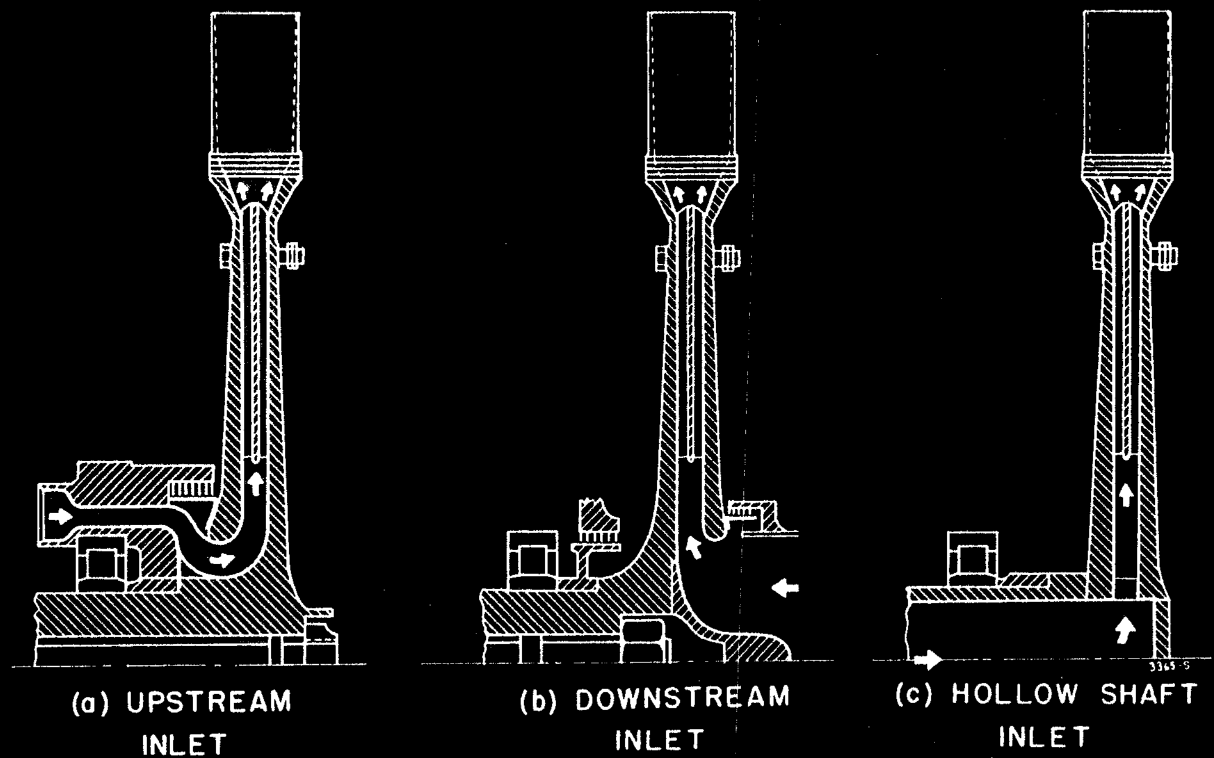

Figure 14. - Split-disk type air-cooled turbine disk configurations.

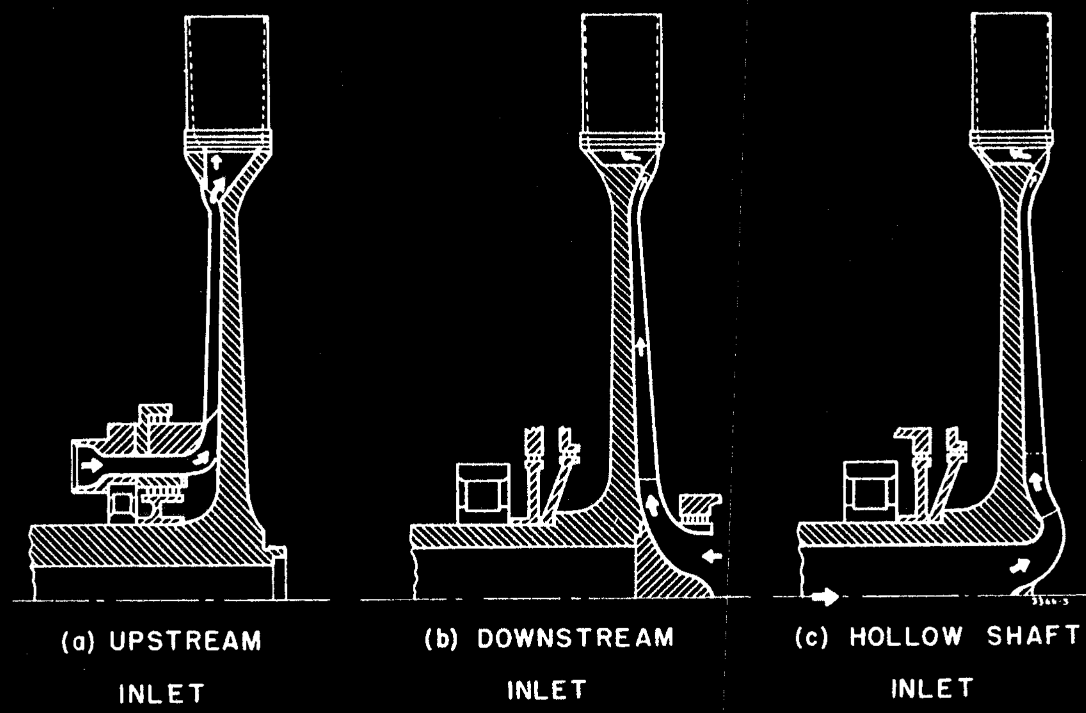

Figure 15. - Shrouded type air-cooled turbine disk configurations.

C33784

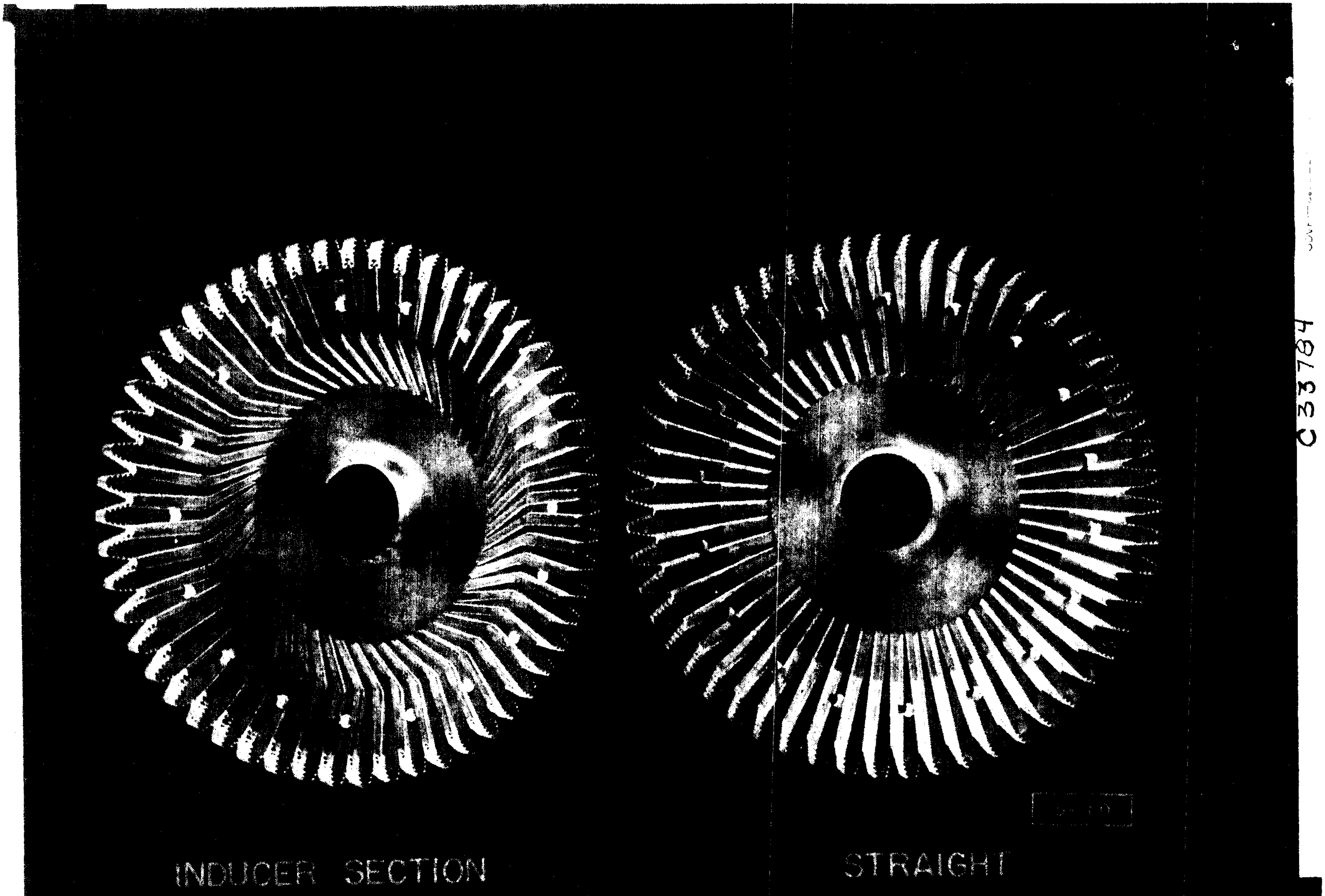

Figure 16. - Air-cooled turbine cooling-air vane configurations.

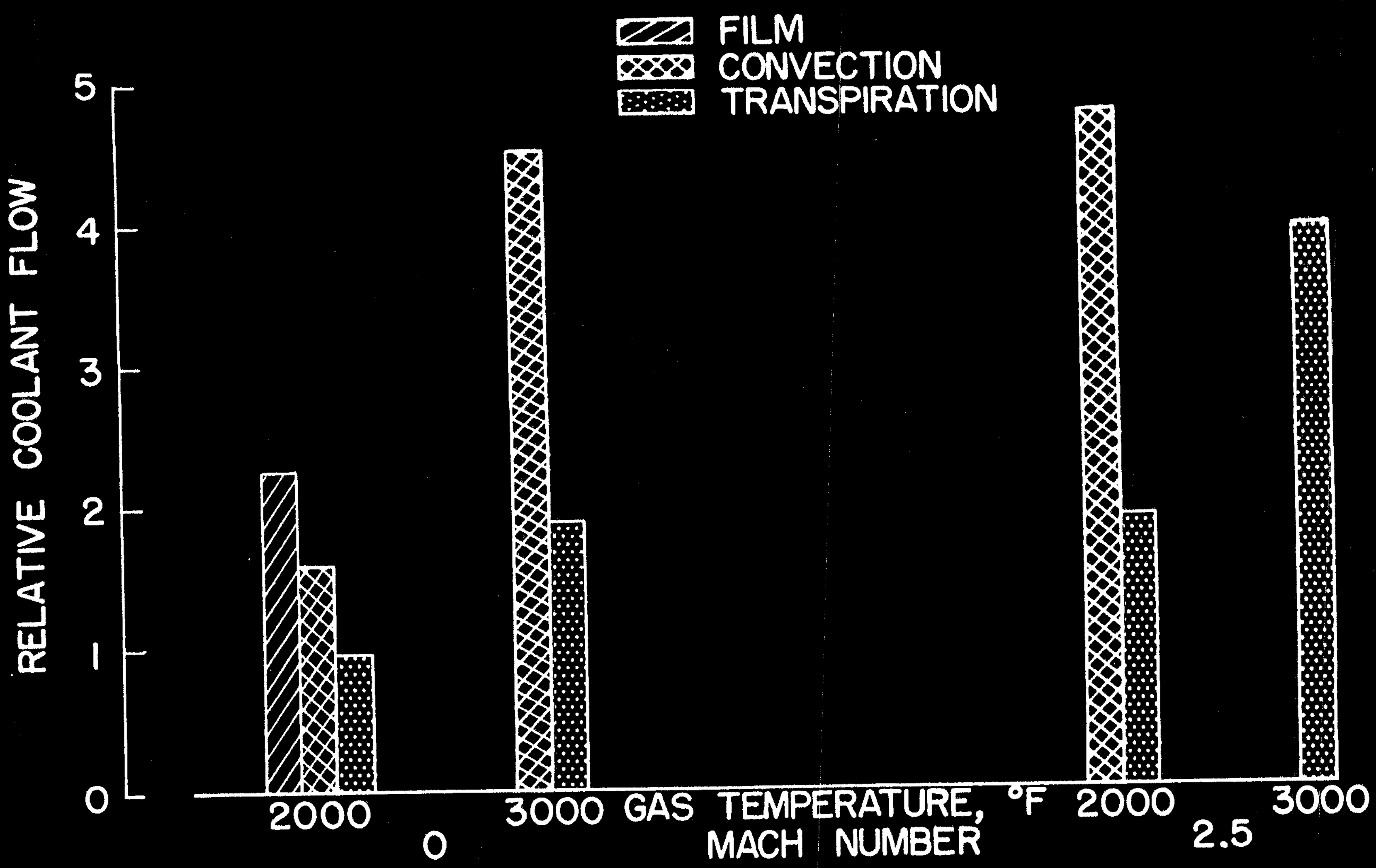

Figure 17. - Comparison of cooling effectiveness of three methods of air-cooling for ranges of turbine inlet temperature and flight Mach number. Cooling air bled from compressor discharge.